United States Patent [19]

McCorkle, Jr.

[11] **3,749,334**

[45] **July 31, 1973**

[54] **ATTITUDE COMPENSATING MISSILE SYSTEM**

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[73] Assignee: **The United States of America as represented by the Secretary of the Army**, Washington, D.C.

[22] Filed: **Apr. 4, 1966**

[21] Appl. No.: **540,479**

[52] **U.S. Cl.** **244/3.22**

[51] **Int. Cl.** **F42b 15/02**

[58] **Field of Search** 244/3.2, 3.21, 3.22; 162/49

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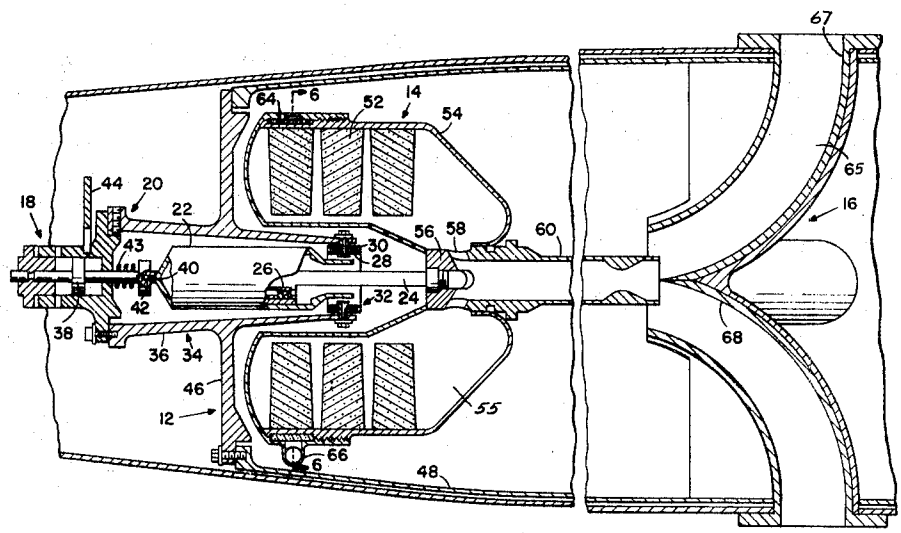
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[57] **ABSTRACT**

A missile utilizing directional control guidance concepts in which torques proportional to missile attitude are generated to correct attitude of the missile during its flight to impact at a target.

16 Claims, 7 Drawing Figures



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SHEET 1 OF 3

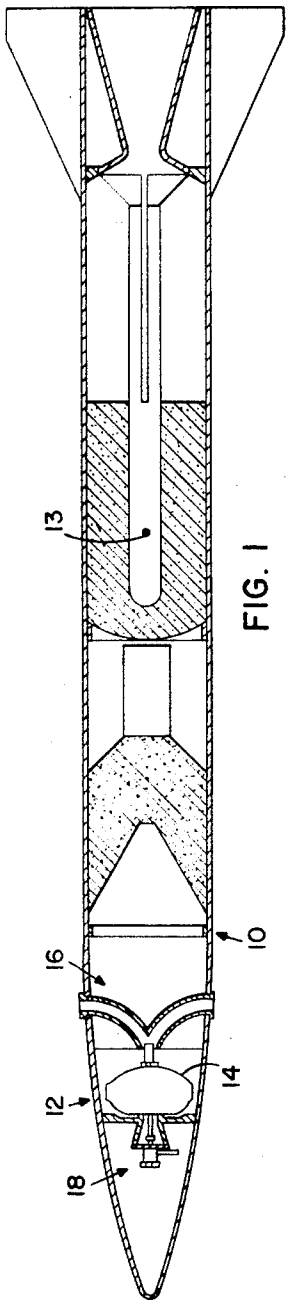


FIG. 1

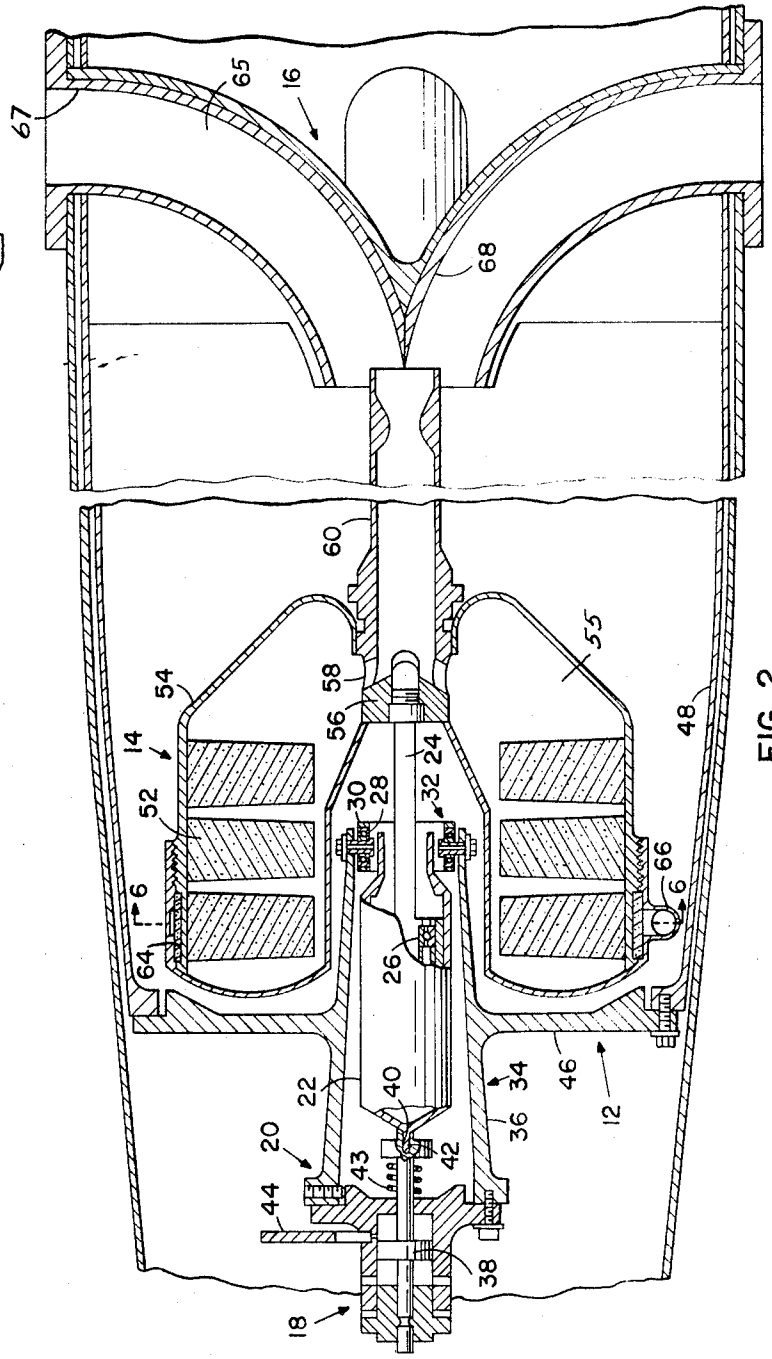


FIG. 2

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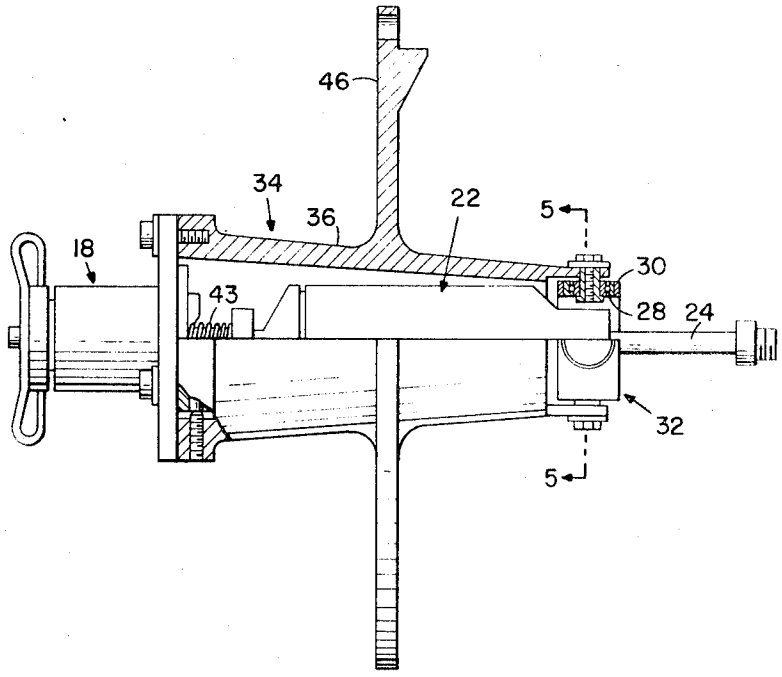


FIG. 3

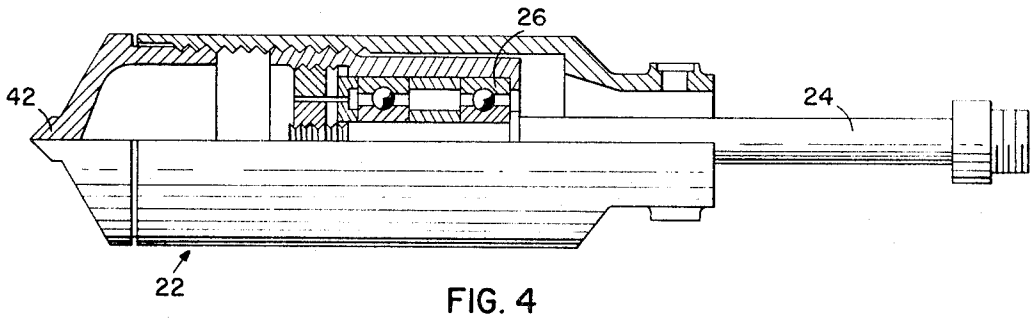


FIG. 4

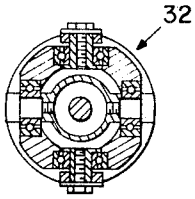


FIG. 5

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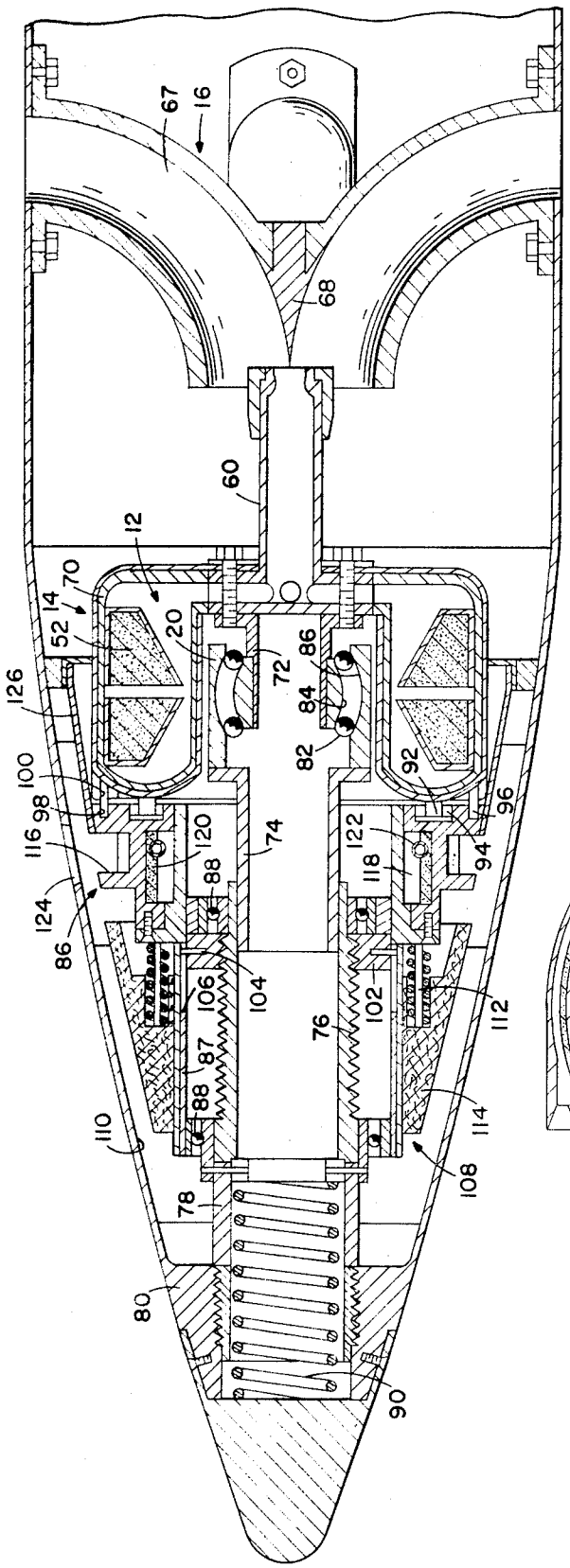


FIG. 7

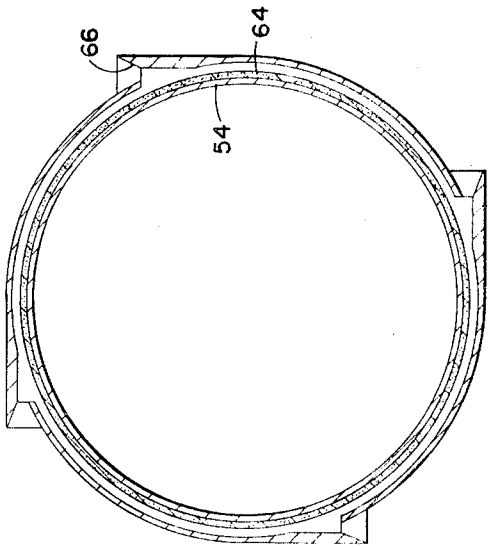


FIG. 6

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ATTITUDE COMPENSATING MISSILE SYSTEM

The device includes a small solid propellant motor as a rotor gimbaled to provide two degrees of freedom (freedom of motion in pitch and yaw) and a gas deflector system to divert the solid propellant motor exit gas flow to provide missile attitude correction side forces. The motor is tailored to the missile acceleration profile to provide a thrust floated gyro which minimizes gimbal bearing loads and thus reduces the gyro drift rate.

Improvement of free rocket accuracy has long been a major objective in weapons research. Rockets are generally superior to cannons in many important areas, such as rate of fire, payload to launcher (or gun) weight ratio, payload to total logistic weight ratio, range capability, and others. The inability to achieve accurate fire with rockets, without the addition of expensive and generally unreliable guidance systems, has been a major drawback and prevented their more general use as artillery.

The quest for accuracy improvement has, in the last few years, included the investigation of simplified guidance schemes. The best known of these schemes at present is the so called directional control concept which utilizes an attitude reference provided by a two degree of freedom gyro to obtain missile attitude information used as an input to a conventional control system which generates corrective torques proportional to missile attitude. The directional control principle is utilized to compensate for the three principle directional error sources, i.e.; (1) linear thrust malalignment or failure of the thrust axis to pass through the rocket center of mass by a certain distance; (2) Mallaunch or the unpredictable component of angular motion about a transverse axis at the instant the rocket leaves the launcher; (3) cross wind effect. Stable pockets turn upwind, and the thrust drives the rocket off the intended path. The cross wind effect is the most difficult error to suppress.

The directional control scheme has been utilized to compensate for the aforementioned sources of error; however, the directional control scheme involves the use of a good gyro, electrical pickoffs, amplifiers, actuators, power supplies and all the evils and complexity these items entail from the standpoint of cost, maintainance and reliability.

The present invention overcomes the undesirable features noted above by providing a system wherein the attitude reference and control system are combined in a single unit thus providing a compact and inexpensive mechanization of the directional control approach which is applicable to a wide range of weapon systems.

The guidance and control system of the present invention includes a two degree of freedom gyro wheel, spun up by a peripheral solid propellant motor. The gyro wheel is provided with a chamber loaded with a solid propellant which is disposed for combustion to provide reaction forces for rotation of the gyro wheel.

Upon ignition of the internal solid propellant charge, mass is ejected through a nozzle affixed to the wheel and aligned with the spin axis thus providing a space stabilized gas jet. The jet is divided into two orthogonal planes, turned through an angle of 90° by a fixed deflector, and exits through four ports located on the missile periphery. These jets provide the missile attitude control forces. As the missile attitude changes, the deflector moves relative to the space stabilized gas jet generating corrective torques proportional to the missile attitude. The mass ejected from the gyro wheel produces a thrust force on the gyro to provide an acceleration thereon.

The present invention provides for accelerations on the gyro, produced by the ejected masses therefrom, which is approximately equal to the accelerations on the missile, and thus, provide a thrust floated gyro which alleviates many of the problems associated with design of gyro gimbals for high "G" applications.

It is an object of the present invention, therefore, to provide a missile having guidance and control components which are incorporated into a single compact unit.

A further object of the present invention is to provide such a guidance and control unit which is "thrust floated" whereby the accelerations acting on the missile and guidance and control unit are equal.

It is a still further object of the present invention to provide a missile with a solid propellant gyro which is rotatable responsive to ignition of the propellant to provide guidance for the missile and to provide control of the missile by utilizing the thrust produced by the propellant of the gyro.

A feature of the present invention provides mounting of the gyro in a manner which allows prealignment of the gyro's dynamic axis so as to greatly reduce the uncaging transient due to mass unbalance of the wheel.

Further objects, features, and advantages of the present invention will become more readily apparent from the following description, taken in conjunction with the accompanying drawings in which:

FIG. 1 is an elevational view, partially in section of a missile utilizing the principles of my invention.

FIG. 2 is an enlarged elevational view, partially in section, of the rocket attitude reference and controller of the present invention.

FIG. 3 is an enlarged elevational view, partially in section, of the gimbals and support structure of the device illustrated in FIG. 2, with the solid propellant and casing removed for clarity.

FIG. 4 is an enlarged elevational view, partially in section of the gimbal and bearing housing of the gyro of the present invention.

FIG. 5 is a view along lines 5—5 of FIG. 3 illustrating the gimbal arrangement of the gyro assembly.

FIG. 6 is a view along lines 6—6 of FIG. 2 illustrating the gyro spinup nozzle, the gyro propellant, casing, bearing housing and support structure being removed for clarity.

FIG. 7 is an elevational view, partially in section, of another embodiment of the present invention.

Referring to the drawings and particularly to FIG. 1, a missile 10 is provided with an attitude reference and controlling mechanism 12 disposed forward of the center of gravity 13 of the missile. The attitude reference and controller mechanism 12 includes a small solid propellant motor 14 as a rotor gimbaled to provide two degrees of freedom (freedom of motion in pitch and yaw) and a gas deflector system 16 to divert the solid propellant exit gas flow to provide missile attitude correction side forces. A caging mechanism 18 is connected to the gyro at the forward end 20 thereof for uncaging or release of the gyro to permit unrestrained rotation thereof.

As shown in FIGS. 2, 3, 4 and 5, mechanism 12 includes a gimbal housing 22 having a rotor shaft 24 rotatably secured in a bearing assembly 26 carried in

housing 22. The rotor shaft extends from the bearing assembly for rotational support in high speed ball bearings 28 carried in a sleeve 30 mounted to a standard two-degree of freedom ball bearing gimbal ring structure 32. Gimbal ring 32 and the bearings are supported in a gyro support structure 34 which serves to mount the entire gyroscope assembly in the forward part of the missile structure. Support structure 34 includes an annular member 36 having gimbal ring structure 32 mounted adjacent the rearward end thereof. The forward end of member 34 supports caging mechanism 18. The caging mechanism (FIG. 2) includes a slidable piston 38 having a conical seat 40 thereon for support therein of an extension of a caging pin 42 of gimbal housing 22. Piston 38 is spring loaded by a compression spring 43 for engagement of the piston with caging pin 22 to firmly lock the gimbal mechanism in place. A firing squib 44 is mounted in caging mechanism 18 for actuation thereof to permit uncaging of the gyro.

The complete assembly is rigidly mounted in the missile by means of a flange 46 secured to member 36 and extending outwardly therefrom and secured to an outer annular member 48 carried by the missile frame.

Solid propellant motor 14 is mounted peripherally about member 36 and includes a propellant 52 carried in chamber 55 of a toroidal casing 54 which is secured to the rotor shaft at 56. Casing 54 is provided with a plurality of passages 58 which communicate into the interior of a nozzle assembly 60 secured to the rearward end of casing 54. A charge of propellant 64 is carried peripherally about casing 54 and provided with suitably canted tangential nozzles 66 arranged to form a Hero's engine (FIGS. 1 & 6) to set rotor shaft 24 in motion. The propellant is carried in an annular internally threaded member 63 which is screwed to the forward end of casing 54.

To divert the gases from motor 14 for utilization as control forces on the missile, a deflector device 16 is rigidly secured to the missile frame and includes a plurality of nozzles or channels 65 communicating with ports 67 opening through the sides of the missile to the atmosphere. Typically, the deflector device includes four channels or nozzles 65 substantially equally spaced and extending radially to communicate with the atmosphere through ports 67. The channels are divided by a splitter 68 disposed on the centerline of nozzle assembly 60. With the gyro aligned with the missile axis, the flow is evenly divided into the four separate channels. It is to be understood that the provision of four nozzles is only typical or illustrative and that a plurality of nozzles may be used. It is necessary, that at least three nozzles be used to provide the restoring torques.

In operation, spin-up propellant 64 and gyro propellant 52 are ignited simultaneously from an external squib (not shown) and the fire is carried to the ignitor material by a mild detonating fuze. Immediately after burnout of the spin-out propellant, squib 44 of caging mechanism 18 is ignited and piston 38 is pressurized by gas fired by the squib and moved rapidly from pin 42. A spring loaded detent 43 holds piston 38 away from pin 42 responsive to movement of the piston, for uncaging of the gyro.

Just prior to uncaging, propellant charge 52 is ignited. The exhaust from this combustion is carried through passages 58 to nozzle assembly 60 to provide a thrust which would, if the rotor were free to move, provide the entire mass carried on gimbal ring structure 32, and including approximately one-half of the gimbal ring mass, an acceleration approximately equal to the missile acceleration. The purpose of such an acceleration applied to the rotor and gimbal-borne structure is to minimize the effects of gimbal imperfections and malalignments, as well as to reduce the gimbal friction by effectively unloading the gimbal bearings. The term "thrust-floated" is thus used to describe this phenomenon when obtained by the reaction of gases discharged from the rotor through a nozzle concentric with the rotor spin axis. In the usual case (non thrust-floated), the center of mass of the gimbal borne structure must lie on both gimbal axes (assumed orthogonal and in the same plane). The rotor spin axis is, in this type of gyroscope, always orthogonal to the plane determined by the gimbal axes, called the gimbal plane. Location of the center of mass along the spin axis relative to the gimbal plane is not critical due to the assumed direction of missile acceleration relative to the spin axis, (i.e., essentially along the spin axis) but is critical with respect to center of mass location in the gimbal plane. A deviation of the center of mass in the gimbal plane from the point determined by the intersection of the two gimbal axes (gimbal center), produces, under the assumed missile acceleration condition, a precessing torque proportional to the acceleration, the gimbal-borne mass, and the deviation distance of the center of mass from the gimbal center. Such precession is highly undesirable since the reference axis established by the gyro changes, instead of remaining fixed in space as intended. The thrust-floated gyroscope presented here, by removing the disturbing torque to the extent that the thrust to gimballed mass ratio matches missile acceleration, either permits greater tolerance on center of mass-gimbal center alignment, or greater accuracy (less drift) for a given set of manufacturing tolerances. Also, and equally as important, the unloading of the gimbal bearings achieved by thrust-floating greatly reduces friction, which, in the presence of gimbal motion (due to missile angular motion), is a significant source of disturbing torques.

The flow from nozzle 60 impinges on a four-way deflector which divides the flow into four streams which are then turned 90° out from the missile axis and exit through the ports in the missile skin placed 90° around the missile. When the nozzle is centered on the deflector axis, the flow is evenly divided among the four streams, and no net force on the missile results. However, when the missile moves transaxially to the rotor spin axis, the flow is divided unequally to provide a variable flow through the nozzles of the deflector assembly, and a net reaction force appears to drive the missile back into alignment with the gyro axis responsive to movement of the missile relative to the rotor, as required for the directional ontrol guidance concept. The above arrangement assumes the exit ports are forward of the missile center of gravity. If the ports are aft of both the missile center of gravity and the aerodynamic center of pressure, the flow passages must be arranged so that flow entering any channel of the deflector exits on the opposite side of the missile from that shown in the drawings and discussed above.

The magnitude of the control force F_c generated in either control plane is given by

$$F_c = T\, l\, f/R\, \phi$$

Where T is the thrust from the rotor, l is distance from

the gimbal center to the deflector entrance (or equivalently, the nozzle exit plane), R the radius of the nozzle exit, f a factor less than unity expressing deflector efficiency, and ϕ the angle between the missile (or deflector axis) and the gyro axis for the control plane in question.

It is to be understood that other spinning mechanisms may be utilized to spin up the gyro, such as a spring motor, an electric motor, or gas jets impinging tangentially on pockets milled into the rotor.

Another embodiment of the present invention is illustrated in FIG. **7** wherein like reference numerals refer to like parts. In this embodiment the spin-up motor is removed from the gyro motor, and spin-up is accomplished by a separate device which simultaneously provides several other functions and features, in addition to spin-up including:

a. Uncaging;
b. Covering of spin-up gas escape ports in the missile after completion of spin-up;
c. Missile axial spin after start of missile forward motion;
d. A relatively uncluttered axial passage through the device, desirable for certain special applications.

As shown in FIG. **7** reference and attitude controller mechanism **12** includes a rotor casing **70** having solid propellant **52** therein forming solid propellant motor **14.** Casing **70** is mounted on a rotor shaft **72** which is rotatably supported in a hollow shaft **74** rigidly affixed to the missile through an externally threaded sleeve or lead screw **76** which, in turn, is secured to a cylindrical member **78** secured to a support member or frame **80** of the missile. Rotor shaft **72,** is mounted for rotation in hollow shaft **74** by means of ball bearing assembly **82** having both inner and outer races **84** and **86,** respectively, provided with a spherical configuration.

To provide for spin-up and uncaging of the gyro, a spin up motor **86** is mounted on ball bearings **88** carried on the periphery of lead screw **76** and is free to move forward thereon. Spin-up motor **86** includes a cylindrical portion **87** enclosing lead screw **76.** The spin-up motor is held initially against rotor casing **70** by a helical spring **90** carried in cylindrical support member **78.** Spring **90** compresses a plurality of soft uncaging flat springs **92** against casing **70.** Springs **92** are carried in grooves **94** provided on the spin-up motor and biased outwardly thereof in engagement with the rotor casing.

Driving connection between the rotor casing and spin-up motor is provided through a plurality of drive pins **96** mounted in apertures **98** and **100** of the motor and rotor casing, respectively.

A lead nut **102** is disposed in threaded relation on lead screw **76** and is moved thereon by means of a plurality of pins **104** projecting through slots **106** provided on cylindrical portion **87** of the spin-up motor. A braking mechanism **108** mounted about cylindrical portion **87** coacts with the inner ogive surface **110** of the missile and a plurality of compression springs **112** to arrest motion of the motor and to impart spin to the missile in a manner described hereinbelow.

As shown in FIG. **7,** the spin-up motor assembly includes the braking mechanism **108** having braking shoes **114** provided with a conical configuration. The braking shoes are rigidly secured to a central motor portion **116** provided with a chamber **118** having propellant **120** therein. A plurality of spin-up nozzles **122** communicate with chamber **118** to carry combustion gases therefrom and provide rotation of the motor. Combustion gases are exhausted to the atmosphere through a plurality of ports **124** disposed around the periphery of the missile. A conical member **126** is secured to central motor portion **116** for slidable movement therewith to seal spin-up exhaust ports **124.**

Operation of the device of this embodiment is initiated upon ignition of spin-up motor propellant **120** and rotor propellant **118,** the gas flow through spin-up nozzles **122** drives the rotor up tp speed. As rotation of the spin-up motor begins, lead nut **102** advances on lead screw **76** until drive pins **104** reach the forward end of slots **106,** uncaging of the gyro commences, and by this time, spin-up propellant **120** is completely consumed. Inertia of the spin-up motor continues to advance lead nut **102** disengaging rotor drive-pins **96,** leaving uncaging flat springs **92** pressing against the rotor. Further advance of lead nut **102** permits springs **92** to reach the limit of their extension, and they leave contact with the rotor, having performed the function of stabilizing the rotor attitude between the time of drive pins **96** disengagement and springs **92** disengagement. Further advance of the lead nut brings spin-up motor conical brake shoe **114** into engagement with missile inner ogival surface **110,** starting compression of springs **112,** thereby arresting motion of the spin-up motor relative to the missile structure, and transferring the spin-up motor angular momentum to the missile causing it to begin axial rotation. The missile propulsion motor is activated just prior to engagement of the brake shoe with the inner missile surface. At completion of spin-up motor braking, the spin exhaust cover member **126** seals the spin exhaust ports **124,** and the control device activation is completed. Gases are expelled through nozzle assembly **16** as described in conjunction with the embodiment described above, for control of missile attitude.

If desired, missile spin could be provided without the use of brake shoe **114** and associated structure. For example, threads on the end of lead screw **76** adjacent cylindrical member **78,** may be blunted so that as lead nut **102** advances on lead screw **76** the nut secured engages the blunted threads of lead screw **76** to impart rotational movement to cylindrical member **78** and the missile frame **80.**

It should be understood that various specific embodiments disclosed are merely illustrative of the general invention and that many modifications thereof may be resorted to that is within the spirit and scope of the present invention.

What is claimed is:

1. A directional control system for maintaining a missile on a launch predetermined path comprising:

a. a gyroscope mounted with its spin axis and rotor aligned with the longitudinal axis of the missile at launch, said missile moveable transaxially relative to said rotor;
b. at least three substantially equally spaced and radially aligned ports mounted on the missile in communication with the atmosphere;
c. a source of pressurized fluid carried by said gyro rotor for providing a reaction force on said rotor which is proportional to the reaction force acting on said missile as a result of the thrust developed therein;
d. a jet nozzle disposed in concentric alignment with said spin axes of said gyro rotor and in communica-

tion with said ports for directing said pressurized fluid along the spin axis of said rotor whereby reaction of the expelled pressurized fluid from said rotor through said concentric nozzle provides a reaction force on said gyro to impart an acceleration thereto substantially equal to the missile acceleration to provide for unrestricted movement of said gyro.

e. said gyroscope having said rotor operatively connected between said source and said ports for providing a variable fluid flow therethrough to provide balancing forces to the missile responsive to its movement relative to said rotor.

2. A missile as in claim **1** wherein said attitude reference device is disposed forwardly of the missile center of gravity.

3. A missile as in claim **2** wherein said gyroscope comprises:

a. a bearing housing secured to the frame of said missile and having a plurality of bearings therein and a rotor shaft mounted in said bearings and extending from said housing;

b. a casing secured to said rotor shaft for rotation therewith and defining said rotor, said casing having a combustion charge mounted therein to form said source of pressurized fluid;

c. actuating means for initiating rotational movement of said casing;

d. means for igniting said combustion charge for rotation of said casing and said rotor shaft; and,

e. means for securing said housing to the frame of said missile in gimballed relation thereto.

4. A missile as in claim **3** wherein said actuating means comprises:

a. an annular member disposed about the periphery of said casing;

b. propellant means carried in a chamber provided in said annular member and disposed for ignition to provide propulsive gases for initial rotation of said casing;

c. a plurality of nozzles disposed peripherally about said annular member and in communication with said chamber to exhaust said propulsive gases therefrom for the rotation of said casing.

5. A missile as in claim **4** including:

a. caging means for securing said housing against movement prior to initiation of rotational movement of said rotor by said actuating means and for release of said housing to permit unrestrained movement thereof subsequent to initiation of said rotational movement by said actuating means; and,

b. means for energizing said caging means for release of said housing.

6. A missile as in claim **1** wherein said gyroscope comprises:

a. A hollow shaft secured to the frame of said missile;

b. a rotor shaft rotatably mounted in said hollow shaft in gimballed relation thereto and extending therefrom;

c. a casing secured to said rotor shaft for rotation therewith and defining said rotor, said casing having said combustion charge therein;

d. drive means releasably secured to said casing for imparting initial rotation thereto, said drive means disposed for release of said casing for unrestrained movement thereof and for imparting spin to the missile for spin stabilization thereof subsequent to release of said casing.

7. A missile as in claim **6** wherein said gyroscope is disposed forwardly of the missile center of gravity.

8. A missile as in claim **7** wherein said drive means comprises:

a. an annular drive member mounted concentrically about said hollow shaft for rotation;

b. actuating means for rotating said annular drive member;

c. braking means carried by said annular drive member for engagement with the frame of said missile for imparting said spin to said missile responsive to rotation of said annular drive member.

9. A missile as in claim **8** including:

a. a support member disposed for support of said hollow shaft and extending therefrom in biased relation with the frame of said missle; and

b. movable means disposed for movement along the length of said support member responsive to rotation of said drive member.

c. means coacting with said movable means and said annular drive member for release thereof from said casing and for moving said brake means into the engagement with the missile frame for imparting spin to said missile.

10. A missile as in claim **9** wherein said actuating means comprises:

a. a combustible charge carried in an annular chamber in said annular drive member and disposed for ignition to provide propulsive gases for rotation of said member;

b. a plurality of nozzles disposed about the periphery of said annular member and in communication with said chamber to exhaust said propulsive gases therefrom for the rotation of said drive member.

11. A missile as in claim **10** wherein

a. said support member is provided with a cylindrical configuration having external screw threads thereon;

b. said means coacting with said movable means includes a nut mounted on said support member in threaded relation therewith;

c. a plurality of pins carried in said nut and extending into slots provided on the inner periphery of said annular drive member for rotation thereby and movement along said support member responsive to rotation of said drive member, said pins disposed for engagement with said annular drive member responsive to a predetermined length of travel of said nut on said support member for movement of said drive member along the longitudinal axis of said missile for release and uncaging of said gyro wheel, said pins disposed for continued movement of said annular drive member until said brake means engages the frame of said missile to impart spin thereto.

12. A missile as in claim **11** including:

a. a plurality of flat springs mounted on an end surface of said annular drive member and in biased engagement with said casing for retention thereof in the caged position;

b. a plurality of pins mounted in slots provided in said surface of said annular drive member for insertion in correspondingly spaced slots provided on said casing to provide the releasable connection be-

tween said drive member and said casing for the initial rotation thereof.

13. A missile as in claim 12 including:

a. a plurality of ports disposed about the periphery of said missile for exhausting propulsive gases of said drive member actuating means to the atmosphere;

b. a member carried by said annular drive member for movement therewith along the longitudinal axis of said missile to seal off said ports at completion of burning of said combustion charge of said drive member.

14. A missile as in claim 13 wherein said control means comprises:

a. a nozzle secured to said casing in axial alignment with the longitudinal axis of said missile and said rotor shaft, said nozzle having a plurality of ports disposed around the periphery thereof in communication with the interior of said casing and nozzle; and,

b. said passage means including a plurality of channels secured to the frame of said missile in communication with the exit of said nozzle and the atmosphere, said channels being concentrically disposed about said nozzle and joining at a point along the longitudinal axis of said missile and said nozzle for equal distribution of said combustion gases through each of said channels when the axes of said reference and said missile coincide.

15. A missile disposed for flight in a trajectory ending in a target comprising:

a. propulsion means carried by said missile for propelling said missile to a target;

b. mechanism carried by said missile for retention thereof in the trajectory, said mechanism including a space oriented attitude reference device having a gyro wheel provided with a combustion charge therein disposed for ignition for providing thrust producing gases for rotating said gyro wheel; and, control means including a plurality of passages communicating with said gyro wheel and the atmosphere for directing said thrust producing gases to the atmosphere through predetermined ones of said passages for providing restoring torques on said missile for restoration of said missile in the trajectory responsive to deviations therefrom;

c. caging means for securing said gyro wheel against movement prior to rotational movement thereof and for release of said gyro wheel to permit unrestrained rotational movement thereof, said caging means including a housing having a piston slidably mounted therein, detent means disposed on said piston for supporting the gyrorotor in a caged position, and means for actuating said piston for movement thereof for release of the gyrorotor to an uncaged position to permit unrestrained movement thereof.

16. A missile as in claim 15 wherein said control means comprises:

a. a nozzle secured to said casing in axial alignment with the longitudinal axis of said missile and said rotor shaft, said nozzle having a plurality of ports disposed around the periphery thereof in communication with the interior of said casing and nozzle; and,

b. said passage means including a plurality of channels secured to the frame of said missile in communication with the exit of said nozzle and the atmosphere, said channels being concentrically disposed about said nozzle and joining at a point along the longitudinal axis of said missile and said nozzle for equal distribution of said combustion gases through each of said channels when the axes of said reference device and said missile coincide.

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